NAVAL POSTGRADUATE SCHOOL
Monterey, California

THESIS

INVESTIGATION INTO THE FEASIBILITY
OF USING SOLID FUEL RAMJETS FOR
HIGH SUPERSONIC/LOW HYPERSONIC
TACTICAL MISSILES

by

Michael A. Witt

June 1989

Thesis Advisor D. W. Netzer

Approved for public release; distribution is unlimited.

Unclassified

security classification of this page

REPORT DOCUMENTATION PAGE

1a Report Security Classification Unclassified	1b Restrictive Markings
2a Security Classification Authority	3 Distribution Availability of Report Approved for public release; distribution is unlimited.
2b Declassification Downgrading Schedule	
4 Performing Organization Report Number(s)	5 Monitoring Organization Report Number(s)
6a Name of Performing Organization Naval Postgraduate School — 6b Office Symbol *(if applicable)* 33	7a Name of Monitoring Organization Naval Postgraduate School
6c Address *(city, state, and ZIP code)* Monterey, CA 93943-5000	7b Address *(city, state, and ZIP code)* Monterey, CA 93943-5000
8a Name of Funding Sponsoring Organization — 8b Office Symbol *(if applicable)*	9 Procurement Instrument Identification Number
8c Address *(city, state, and ZIP code)*	10 Source of Funding Numbers Program Element No / Project No / Task No / Work Unit Accession No

11 Title *(include security classification)* INVESTIGATION INTO THE FEASIBILITY OF USING SOLID FUEL RAMJETS FOR HIGH SUPERSONIC/LOW HYPERSONIC TACTICAL MISSILES

12 Personal Author(s) Michael A. Witt

13a Type of Report	13b Time Covered	14 Date of Report *(year, month, day)*	15 Page Count
Master's Thesis	From To	June 1989	62

16 Supplementary Notation The views expressed in this thesis are those of the author and do not reflect the official policy or position of the Department of Defense or the U.S. Government.

17 Cosati Codes			18 Subject Terms *(continue on reverse if necessary and identify by block number)*
Field	Group	Subgroup	scramjet,dual mode, supersonic combustion

19 Abstract *(continue on reverse if necessary and identify by block number)*

An investigation was conducted to determine the feasibility of using solid fuel ramjets as propulsion units on high supersonic low hypersonic tactical missiles. Experiments were conducted on two types of configurations. Plexiglas was used as the fuel in a scramjet and HTPB was used as the fuel in a dual mode combustor. Results indicated that supersonic combustion occurred in both configurations, but that mixing and heat addition losses were high. Ignition limits were identified as a possible limiting factor in the use of solid fuels for the proposed application. Combustion kinetics were shown to be rapid enough to support sustained combustion in supersonic flow.

20 Distribution Availability of Abstract ☒ unclassified unlimited ☐ same as report ☐ DTIC users	21 Abstract Security Classification Unclassified	
22a Name of Responsible Individual D. W. Netzer	22b Telephone *(include Area code)* (408) 646-2980	22c Office Symbol 67Nt

DD FORM 1473,84 MAR — 83 APR edition may be used until exhausted. All other editions are obsolete — security classification of this page

Unclassified

Approved for public release; distribution is unlimited.

Investigation into the Feasibility of Using Solid Fuel Ramjets for High Supersonic/Low Hypersonic Tactical Missiles

by

Michael A. Witt
Lieutenant Commander, United States Navy
B.S., United States Naval Academy, 1976

Submitted in partial fulfillment of the requirements for the degree of

MASTER OF SCIENCE IN ENGINEERING SCIENCE

from the

NAVAL POSTGRADUATE SCHOOL
June 1989

Author: ______________________
Michael A. Witt

Approved by: ______________________
D. W. Netzer, Thesis Advisor

____
R. P. Shreeve, Second Reader

____
E. R. Wood, Chairman,
Department of Aeronautics

____
G. E. Schacher,
Dean of Science and Engineering

ABSTRACT

An investigation was conducted to determine the feasibility of using solid fuel ramjets as propulsion units on high supersonic/low hypersonic tactical missiles. Experiments were conducted on two types of configurations. Plexiglas was used as the fuel in a scramjet and HTPB was used as the fuel in a dual mode combustor. Results indicated that supersonic combustion occurred in both configurations, but that mixing and heat addition losses were high. Ignition limits were identified as a possible limiting factor in the use of solid fuels for the proposed application. Combustion kinetics were shown to be rapid enough to support sustained combustion in supersonic flow.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

LIST OF SYMBOLS

A	Cross sectional area
A^*	Cross sectional area at choking
D_p	Grain port diameter
f	Fuel to air ratio
g_c	Gravitational constant
G	Mass flux
k	Constant in regression rate equation
L_g	Grain length
m	Mass flow rate
M	Mach number
P	Pressure
r	Fuel regression rate
R	Gas constant
T	Temperature
x	Constant in regression rate equation
y	Constant in regression rate equation
z	Constant in regression rate equation
γ	Specific heat ratio
$\eta_{\Delta T}$	Thermal combustion efficiency
ρ_f	Fuel density
ϕ	Equivalence ratio

Subscripts

a	Air
bp	Bypass
c	Combustion chamber

exp	Experimental
f	Fuel
i	Air inlet
mix	Mixer region
stoich	Stoichiometric
t	Stagnation
th	Theoretical
w	Wall static
1,2,3	Stations defined in Figure 13

ACKNOWLEDGEMENTS

My heartfelt thanks goes to Professor Dave Netzer for the light he shined along the path to understanding, for unending patience, and for his eternal optimism. Thanks also to Harry Conner, who knows just how to handle "knowledgeable" officers, to Pat, John, and model maker Don, without whose help and support this investigation could never have been completed. Finally, to my wife, who stood tough through it all, my love and gratitude.

I. INTRODUCTION

The advent of ultra long range air attack weapons that operate at altitudes and Mach numbers beyond the capability of current anti-air warfare weapons has created a requirement for high supersonic/hypersonic, long range weapons that can effectively engage targets prior to hostile weapon release. Altitudes above 80,000 feet and intercept ranges of 50-100 nautical miles for air launched, and 150-300 nautical miles for surfaced launched weapons may be required to counter this threat. As the abilities of hostile platforms in the areas of maneuverability and counter-detectibility improve, so must the capabilities of the weapon used against them. High speed, highly maneuverable targets must be countered with weapons that have powered flight throughout the engagement, especially during the end game, in order that a high probability of target destruction can be achieved.

The current anti-air weapon of choice is the solid propellant rocket. Examples of such weapons in the inventory include the Standard, Sparrow, and Phoenix missiles. Solid propellant rockets are well suited to the short and medium range engagements because they are easy to handle and store, have high thrust-to-weight ratios, and have constant specific impulse throughout the propellant burn time. They are capable of attaining speeds of Mach 1.5 to Mach 2.5 above launch

speed. The technology base is relatively mature and they are particularly well suited for shipboard use because of the propellant stability during stowage.

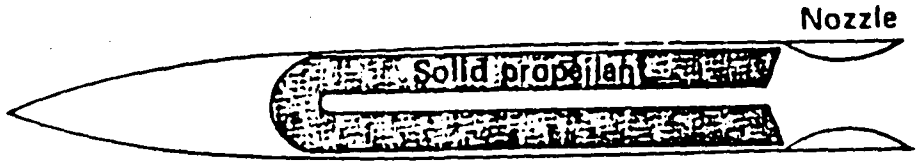
The primary drawback of rockets is that the engine burn time is fairly short, accelerating the weapon to maximum velocity then leaving it to coast to the target intercept. This can result in greatly reduced terminal velocities for long range target intercepts. The reduced velocity available at the end game may prevent the missile from maneuvering sufficiently to make its warhead effective. Pulsed motors and booster/sustainer type motors improve the situation somewhat, but since the rocket must carry both the fuel and the oxidizer, the maximum range will be limited to less than that of a weapon that can fill the entire propellant volume with an equally energetic fuel and gather the oxidizer from the surrounding atmosphere. In order to maximize specific impulse, metals are often used in the rocket propellants. These metal additives may result in significant exhaust plumes and may also increase other signatures that could give the hostile target an early warning of launch or weapon position during flight.

The most thermodynamically efficient mode of propulsion up to speeds of about Mach 3.0 is the gas turbine engine. As speed increases from that point the ram pressure overtakes the

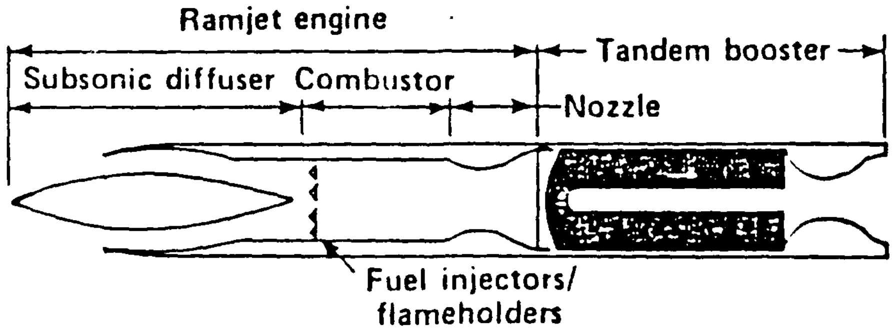
capabilities of the compressor. The structural limits of the materials in the compressor can be exceeded at high Mach numbers and the increased air temperature will limit energy addition in the combustor.

The ramjet is the most energy efficient power plant at speeds from Mach 3.0 to about Mach 5.0 - 7.0, depending on the air inlet type and altitude. Ramjets may be either solid or liquid fueled, but the discussion here will, unless otherwise specified, pertain solely to the solid fueled types. A derivative of the ramjet, the ducted rocket, is also a viable propulsion type for long range applications, but will not be discussed here. Figure 1 shows the basic design features of the propulsion types discussed thus far.

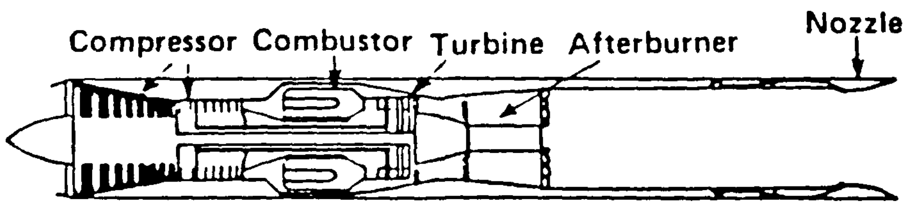
Ramjets depend upon the use of semi-isentropic compression and/or shocks in the inlet diffuser to reduce the velocity of the incoming combustion air to subsonic speeds and raise its static temperature so that combustion can be sustained. Inlets often utilize a series of oblique shocks that gradually reduce the speed of the air before using a normal shock to accomplish the subsonic conversion. Since the stagnation pressure loss across a shock is an irreversible process, it is desireable to make the shocks as weak as possible so that the performance of the engine can be at a maximum. The flow through the combustor must be at low subsonic speeds to allow adequate time for mixing and complete burning of the fuel. If the flow of the air is too rapid, poor combustion, or in



(a) Solid-fueled rocket



(b) Integral rocket ramjet



(c) Turbojet

Figure 1 - Propulsion systems for tactical missiles: (a) solid propellant rocket; (b) integral rocket ramjet; (c) turbojet. [Adapted from Ref. 1: p. 139]

the limit, flame blowoff can occur. The flow within the combustion chamber ranged from Mach 0.2 to Mach 0.4. In order to counter the effects of variations in the air velocities, to stabilize the flame, and reduce the chance of blowoff, flame holders are used. In solid fuel ramjets the most common flame holders are rearward facing steps at the combustor inlet. These provide a low velocity, fuel rich recirculation zone that promotes fuel mixing and combustion. The gases in the combustor are raised to temperatures near 2500 K in the burning process and then are expelled through a converging-diverging nozzle to maximize exit velocity and thrust. Figure 2 depicts a typical solid fuel ramjet configuration and combustor flow pattern. [Ref. 2]

There are some requirements for ramjet operation that, until recently, have resulted in limited use of that propulsion type and that severely restrict their use at Mach numbers above Mach 6.0. At speeds less than approximately Mach 0.9, the ram pressure is too low and the total drag on the missile overcomes the thrust generated by the engine. At high Mach numbers the air temperatures limit heat addition and structural material limits may be exceeded. In addition, due to the wide range of engine size design requirements between launch and cruise conditions, an external rocket booster was required to accelerate the ramjet to operating speed. This resulted in very large propulsion packages like those in the

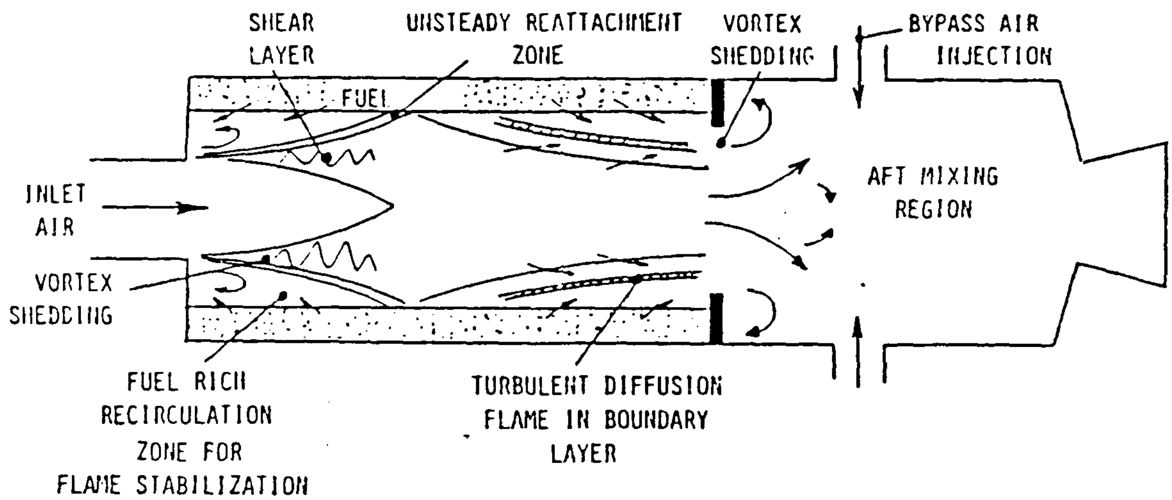


Figure 2 - Solid fuel ramjet engine [Ref. 2: p. 16]

TALOS and BOMARC missiles. The development of the integral rocket ramjet has relieved the separate booster requirement, making more moderately sized weapons feasible. In an integral rocket ramjet the booster propellant is contained in the combustion chamber of the ramjet. Frangible covers provide a means to open the air inlets when the missile is up to starting speed for the air breathing engine. Often, an ejectable nozzle is also required to meet the sizing needs of the ramjet. Figure 3 depicts a typical integral rocket ramjet configuration. Figure 4 illustrates a typical performance envelope for ramjets.

In order to limit the static temperatures at the combustor inlet at high Mach numbers, the flow into the combustor must remain supersonic. This is the definition of a supersonic combustion ramjet (scramjet). A comparison of the flight performance ranges of the gas turbine, ramjet,scramjet, and rocket is shown in Figure 5.

There is a great deal of research being directed at the area of supersonic combustion. The majority of the research has been related to the development of the National Aerospace Plane (NASP). The NASP research has centered on liquid and gaseous fuels, witn hydrogen or mixtures of hydrogen and hydrocarbons as the primary fuel types.

There are two basic types of scramjet configurations. In the first type all the incoming combustion air is mixed directly with the fuel while travelling at speeds greater than

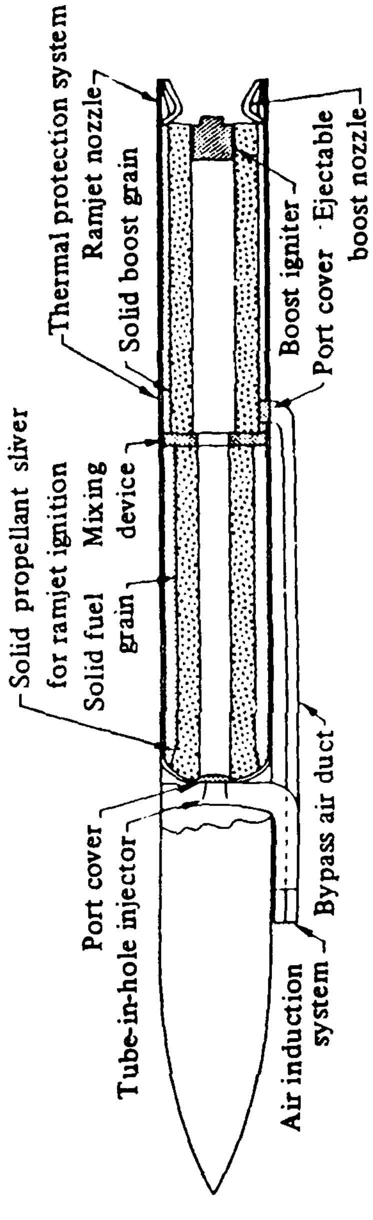


Figure 3 - Integral rocket ramjet [Adapted from Ref. 3: p. 28]

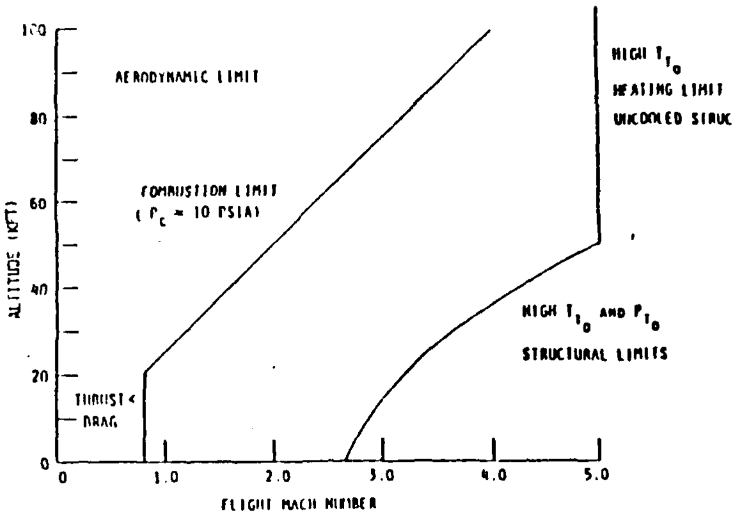


Figure 4 - Performance range of ramjets [Adapted from Ref. 4]

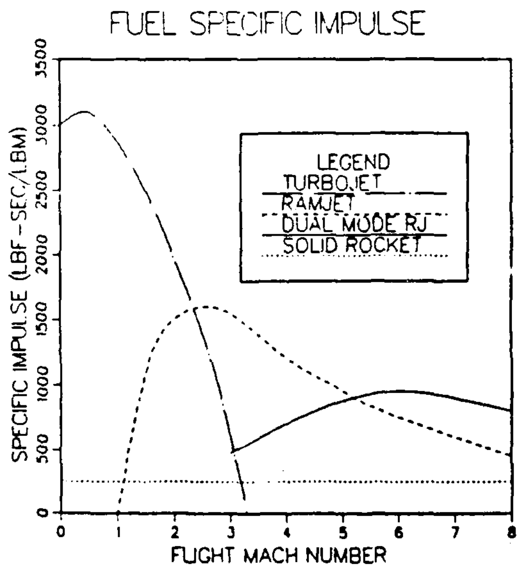
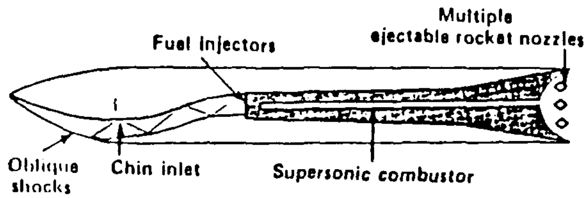
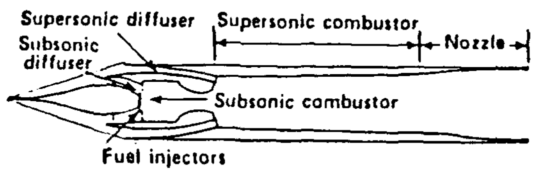


Figure 5 - Theoretical performance envelopes. [Adapted from Ref. 1: p. 144]

Mach 1.0. For the remainder of this discussion, this configuration will be referred to as the scramjet configuration. In the second type, a portion of the combustion air is shocked down to subsonic speeds for combustion in a standard ramjet operating at a very fuel rich condition, providing the fuel rich exhaust to be combined with the remaining bypassed supersonic combustion air for burning in a supersonic combustor. Since the second type utilizes two modes of combustion it is called a dual mode ramjet (DMRJ). Diagrams of the two scramjet types are shown in Figure 6.



(a) Integral rocket liquid-fueled scramjet



(b) Liquid-fueled dual-combustor ramjet.

Figure 6 - Supersonic combustion systems for tactical missiles. [Adapted from Ref. 1 : p. 142]

The fuel flow in a solid fuel ramjet is dependent upon the flow rate and static temperature of the incoming air. These parameters determine the heat flux to the fuel surface and thus determine the rate at which the fuel is vaporized off the

wall of the grain. The average vaporization rate, or regression rate, occurs generally in accordance with the equation

$$r = k\ G^x\ p^y\ T^z$$

where G is the mass flux entering the combustor, p is the chamber pressure, T is the inlet air static temperature, and k,x,y, and z are constants specific to the fuel type. Examination of the regression rate equation reveals some of the problems associated with using solid fuels for scramjet applications. For a given flight condition (altitude, Mach number, and mass flow rate), the regression rate will be lower for the scramjet than for the ramjet because of the lower combustor pressures and static air inlet temperatures in the scramjet. Once evaporated, the fuel and air must be adequately mixed in a manner that allows efficient combustion without incurring large losses in stagnation pressure.

Several factors can affect or control the combustion process for solid fuels in supersonic flows. First is residence time and its relationship to the kinetic rates of the reactants after they have been adequately mixed. Sufficient combustor length must be provided to allow complete combustion of the reactants. Second, heat transfer from the air must be sufficient to provide the necessary amount of fuel

vaporization. Third, and perhaps the most important, is whether proper mixing and flame holding can be provided.

With respect to chemical kinetics, analytical studies by Billig, et. al. [Ref. 5] and Vaught, et. al. [Ref. 6] have shown that, with adequate mixing, the kinetics should be rapid enough to remove that concern from the operation of the DMRJ. For the scramjet, however, kinetics could be a limiting factor for solid fuels. In the DMRJ, the subsonic combustor is used to process the slower reactions, leaving only the very rapid reactions like

$$2H + .5O_2 \text{ ---> } H_2O \quad \text{and}$$

$$CO + .5O_2 \text{ ---> } CO_2$$

for the supersonic part of the combustor. The scramjet does not have the benefit of the time to break down the complex fuel molecules at a slower pace, and, therefore, may not accomplish efficient combustion in realistic combustor lengths.

Adequate fuel vaporization and mixing is not a concern for the subsonic portion of the DMRJ combustor, but may be critical in the case of the scramjet. Mixing has been proven to be poor in supersonic flows and complete combustion may be difficult to obtain in either combustor type [Ref. 7]. As the Mach number increases, the boundary layer, where most of the mixing and combustion takes place, decreases in thickness. With the diminishing mixing layer comes a decrease in the flame holding capacity of the burner. As in the ramjet, the

most common flame holding design in scramjets is the rearward facing step. The sudden expansion caused by the step separates the boundary layer which results in an expanding shear layer and generates shock waves that may further enhance mixing, but may also induce large losses. There has been some research into the use of shocks to aid in mixing and flame stabilization, but it has been directed in the area of gaseous or liquid fuel injection type systems.

The purpose of this study was to investigate the feasibility of using a solid fuel for a high supersonic/ low hypersonic tactical missile in a dual mode and/or supersonic combustor configuration. The dual mode combustor utilized HTPB fuel for the subsonic gas generator, and Plexiglas was used to investigate supersonic combustion in the grain.

II. DESCRIPTION OF THE APPARATUS

The test facility consisted of a high pressure air supply with two test stands. Each test stand was equipped with a methane fueled air heater. The exhaust ducting of the air heater limited the air heater temperature to approximately 1650°R. This restricted the high altitude Mach number conditions which could be simulated to approximately Mach 4.0.

HTPB has been widely used for solid fuel ramjet studies. Because of its well known behavior it was selected for use in the DMRJ investigation. For supersonic combustion in the grain it was necessary to investigate varying geometries in order to study mixing and flame holding characteristics. For this reason Plexiglas was selected as the fuel for its ease of fabrication.

Data acquisition was accomplished using a Hewlett-Packard 3054A Data Acquisition System in conjunction with a Hewlett-Packard 9836 computer. A complete set of point data was taken every 1.5 seconds for the DMRJ. Data was collected for the scramjet at 0.5 second intervals. The low data rate for the DMRJ was driven by the number of readings taken per cycle and the speed of the 3054A processor. The pressure-time relationships for combustion pressure in the SFRJ combustor, the supersonic combustor static exit pressure, and the

supersonic combustor exit stagnation pressure were recorded on a strip chart recorder to provide continuous data throughout the burn time.

A. SCRAMJET APPARATUS

The test conditions for the solid fuel SCRAMJET were as follows:

a. Simulated flight: M_0=4.0, h=80,000 ft.

b. Air mass flow rate: 0.5 lbm/sec.

c. Combustor inlet Mach number: 1.5

d. Combustor grain geometry: various (to examine mixing rates and flame stabilization)

e. Fuel: Plexiglas (ρ=0.0426 lbm/in^3)

f. Fuel for ignition and pilot as required: hydrogen

g. Heater fuel: methane

h. Ignitor fuel: ethylene

A schematic of the test apparatus is shown in Figure 7. Vitiated, heated air was injected into the combustor through a converging-diverging nozzle designed to provide a flow Mach number of approximately 1.5. Initial screening tests were conducted to evaluate various burn patterns, flame stability, and mixing rates in different grain geometries. These tests resulted in the selection of two grain geometries for further study. Diagrams of the selected grains are shown in Figure 8. Photographs of the apparatus are given in Appendix B.

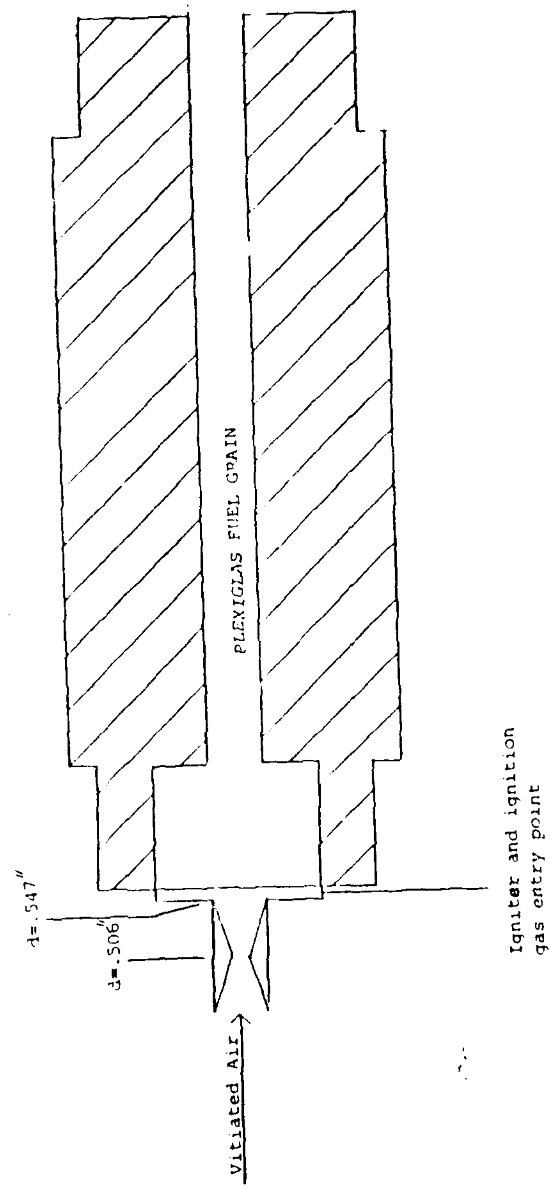
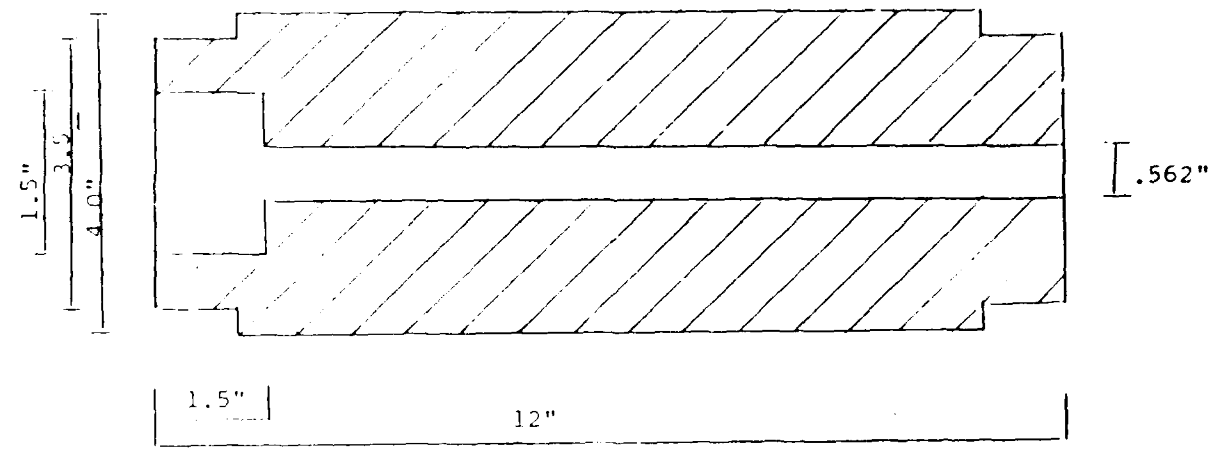
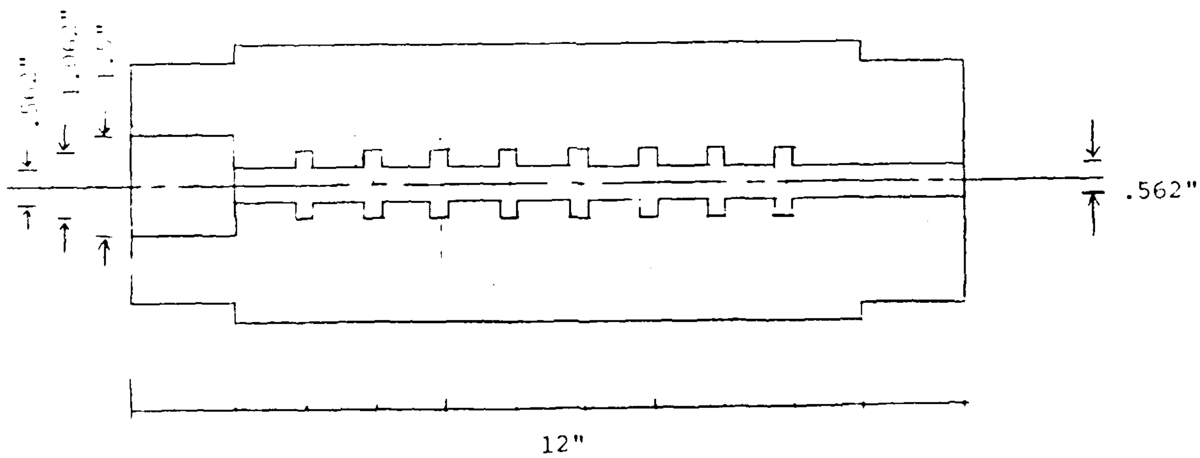


Figure 7 - Schematic of scramjet test apparatus.



a. Smooth bore test grain.



b. Grooved bore test grain.

Figure 8 - Schematic of scramjet fuel grains; (a) smooth bore; (b) grooved.

Gas pressures and the air heater exit sonic choke were set to provide an air heater static pressure of about 250 psia to ensure efficient and complete combustion of the methane in the air heater. The required fuel-to-air ratio for the air heater was determined using the Naval Weapons Center (NWC) Propellant Evaluation Program, PCPEP. PCPEP calculates the combustion of user prescribed ingredients under equilibrium, adiabatic conditions and provides the theoretical specific heat ratio, molecular weight of the product mixture, combustion temperature, molecular species in the combustor and in the exhaust, and other parameters of possible interest. The combustor pressure and masses (or proportions) of the ingredients are input by the user. To replace the oxygen consumed from the air in the heater, the mass flow rate required was computed by utilizing the following chemical reaction:

$$CH_4 + 2O_2 \dashrightarrow CO_2 + 2H_2O.$$

The molecular weight of methane (CH_4) is 16 gm/mole and the molecular weight of O_2 is 32 gm/mole. Thus, it is necessary to add the oxygen at a mass flow rate four times that of the methane if complete combustion in the heater is assumed.

Once the desired flow rates were established, the sonic choke sizes to be used for metering the flow were established through use of the continuity equation

$$(\text{eqn. } 1)\quad m = \frac{P_t A \sqrt{\frac{\gamma g_c}{R T_t}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}}{A/A^{*}\Big|_{M}}$$

The recirculation zone at the inlet of a Plexiglas grain in a subsonic ramjet application is normally established by using a h/d ratio of approximately 0.33. The grains made for this experiment were also designed to meet the 0.33 inlet ratio, although it was not known if there would be sufficient mixing in the zone or whether there would be enough oxygen in the zone to maintain and/or hold the flame.

The grain bore was sized to produce a flow of Mach 1.6 without heat addition. Both grains were bored to the same diameter. One combustor was a straight, smooth bore and the other was grooved with rings approximately 0.25 in. wide and 0.25 in. deep to enhance mixing and promote flame stabilization in the first half of the grain, then opening to a constant diameter of two inches for the remainder of the grain length.

During the preliminary testing it was discovered that a pilot flame would be required. Due to its excellent clean burning characteristics and proven flame stability in supersonic applications, hydrogen was selected for the pilot fuel. A hydrogen flow of 0.08 percent of the airflow proved sufficient to establish a stable flame through the range of velocities experienced as the grain bore widened during the burn.

Pressure taps were installed at the head and exit areas of the grain in order that the static pressures in the combustor could be monitored. The grain exhausted directly to the atmosphere, without an exit nozzle. At the exit of the grain a pitot tube was installed to gather data on the stagnation pressure at the combustor exit in the area of the exit static pressure tap.

Since the Plexiglas fuel was nearly transparent, a video recording system was used to aid in determining the burn characteristics of the combustor. The presence of shocks and the effectiveness of the flameholders could be readily evaluated visually.

B. DMRJ APPARATUS

The test conditions for the DMRJ were as follows:

a. Simulated flight: $M_0 = 4.0$, h = 80,000 ft.

b. Air mass flow rate: 1.0 lbm/sec.

c. Subsonic / supersonic air flow fractions: 50/50.

d. Fuel: HTPB ($\rho = 0.0332$ lbm/in^3).

e. Equivalence ratio (gas generator): $\phi_{SFRJ} = 1.9$

f. Equivalence ratio (overall): $\phi_{total} = 0.95$

g. Flow velocity in supersonic combustor: Mach 2.5

h. Air heater fuel: methane

i. SFRJ igniter fuel: ethylene

A schematic of the apparatus is given in Figure 9. Two air heaters were used to supply the combustion air, using the same

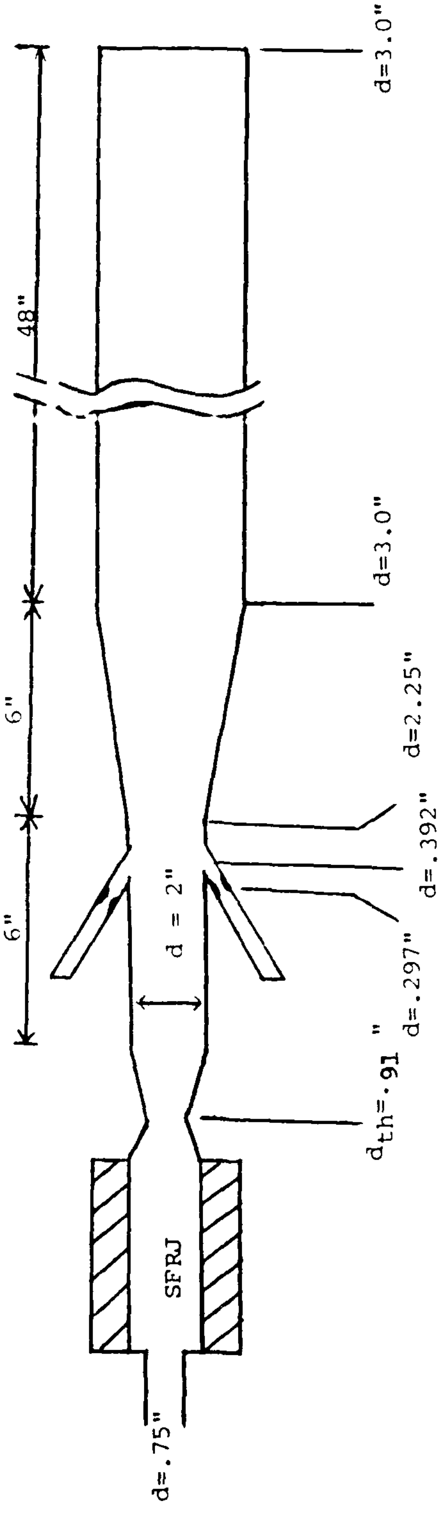


Figure 9 - Schematic of DMRJ test apparatus.

method as previously described for the SCRAMJET. The fuel rich exhaust of the SFRJ section exhausted into the supersonic combustor at a Mach number of approximately 2.7 where it was mixed with the bypass air entering the mixing region at approximately Mach 1.5 - 1.6. The bypass air was injected at an angle of 30 degrees from the combustor centerline through two ports on opposite sides of the combustor into a two inch long constant area mixing chamber. After mixing, the flow proceeded through an expansion/transition section to a constant diameter tube exhausting to the atmosphere.

A pitot tube arrangement, as in the SCRAMJET, was used to collect data on the exit stagnation pressure. Pressure taps were also available along the length of the combustor to allow investigation of shock locations. Thermocouples were placed at the bypass air inlet upstream of the inlet nozzle, at the SFRJ exhaust, at the mixing area, and at the inlet of the SFRJ to monitor conditions in the areas of interest.

The length of the fuel grain required to produce the Φ_{SFRJ} listed above was computed using the regression rate equation with the following values:

a. $k = 7.1789 \times 10^{-5}$

b. $x = 0.53$

c. $y = 0.33$

d. $z = 0.71$

in conjunction with the equations

$$m_f = \rho_f \pi D_p L_g r,$$

$$f = m_f/m_a,$$

and $$\Phi = f/f_{stoich} = f/0.0734.$$

The SFRJ combustor section consisted of a constant diameter (nominally 1.75 in.) grain port with a 0.75 in. diameter air inlet and a converging-diverging nozzle with a throat diameter of 0.91 inches. With a grain length of 13 in., a desired combustor pressure of 115 psia, an air mass flow rate of 0.45 lbm/sec, and an inlet static temperature of 1640 R, a ϕ_{SFRJ} of 1.95 was predicted.

Photographs of the apparatus are given in Appendix B.

III. EXPERIMENTAL PROCEDURES AND ANALYSIS

The procedure for the experiment was as follows:

a. Compute desired flow rates and install sonic chokes.

b. Calibrate transducers, weigh and measure fuel grain.

c. Set flow rates for air, heater gases, pilot/ignition gas, and igniters.

d. Verify proper igniter operation.

e. Verify heater operation.

f. Start SCRAMJET (or DMRJ) computer program.

g. Set times for heating, ignition, burn, and purge.

h. Start the air heater and monitor temperature.

i. Commence the test. Secure all gases when complete.

j. Re-weigh the grain.

k. Calculate m_f and ϕ.

l. Calculate T_{t4th} based on m_f,m_{air},and p_c(average) using PCPEP.

m. Calculate T_{t4exp} using the pitot stagnation/static pressure relationships.

n. Calculate $\eta_{\Delta T}$ using T_{t4exp}, T_{t4th} and T_{ti}.

o. Analyze the video recording.(SCRAMJET only)

The computer program SCRAMJET was written to control the experiment by computer once the air heater operation was established. It routed the combustion air through the motor, scheduled the collection of data, controlled all gas flows and igniter electrical systems, and conducted the purge to extinguish the burn. It also calculated the measured flow rates and temperatures from the data collected and presented the results along with all preset parameters for comparison.

The computer program DMRJ was written to accomplish the same goals for the dual mode tests that SCRAMJET did above. The DMRJ data collection system required many more measurements than did the SCRAMJET system, driving the data acquisition rate down to 40 data sets per minute as compared to 120 data sets per minute in the SCRAMJET program. In order to prevent the slow data rate from restricting observation of performance, the SFRJ combustor pressure, supersonic combustor exit static pressure, and exit stagnation pressure were recorded on strip charts.

A. SCRAMJET ANALYSIS

One of the goals of the experiment was to determine the thermal combustion efficiency ($\eta_{\Delta T}$) of the solid fuel scramjet. In order to determine $\eta_{\Delta T}$ several procedures had to be accomplished. The expression defining $\eta_{\Delta T}$ is

$$\eta_{\Delta T} = (T_{t4exp}-T_{ti})/(T_{t4th}-T_{ti})$$

where T_{t4exp} is the experimentally determined stagnation temperature, T_{t4th} is the theoretical stagnation temperature predicted by using PCPEP, and T_{ti} is the inlet stagnation temperature of the combustion air. Of the three required elements, only T_{ti} can be measured directly. The others must be derived. T_{t4exp} is calculated by assuming isentropic flow and using the continuity equation, (eqn. 1). P_t is known from the stagnation probe, the area of the exit is known, and the mass flow rate is the sum of the measured gas and fuel flows. The specific heat ratio, γ, and the gas constant, R, are obtained from PCPEP when run under the measured conditions.

In order to determine the exit Mach number, the static and stagnation pressures measured at the exit are expressed as a ratio resulting in an expression dependent on the exit Mach number and γ only. The expression can be reduced to the form

$$\text{(eqn. 2)} \quad \frac{P_{t2}}{P_1} = \left(\frac{(\gamma+1)M_1^2}{2}\right)^{(\gamma/(\gamma-1))} \left(\frac{2\gamma M_1^2}{(\gamma-1)} - \frac{(\gamma-1)}{(\gamma+1)}\right)^{1/(\gamma-1)}$$

The Mach number can be determined by iteration of this equation. Once the Mach number is determined, the area ratio can be determined using the isentropic formula

$$A/A^* = (1/M_1)((1+((\gamma-1)/2)M_1^2)/((\gamma+1)/2))^{(\gamma+1)/2(\gamma-1)}.$$

Then T_{t4exp} can be calculated. Since T_{t4th} is a product of the same PCPEP analysis that produced γ and R, $\eta_{\Delta T}$ can now be solved.

PCPEP was solved three times for comparison of the conditions where (1) only the hydrogen burned, (2) only the Plexiglas burned and (3) both hydrogen and Plexiglas burned. In this way an indication could be obtained as to whether the grain was being consumed and contributing to the thrust of the motor, or whether the evaporated fuel was merely being carried along the wall in the shear layer and out the end of the motor. Printouts of data are given in Appendix A.

B. DMRJ ANALYSIS

The analysis for the case of the DMRJ was to be similar to that of the scramjet if supersonic flow was maintained in the presence of mixing and heat addition.

IV EXPERIMENTAL RESULTS AND DATA REDUCTION

A. SCRAMJET RESULTS

1. Smooth Bore Test Results

Four tests were conducted using the smooth bore grain, and two tests using the grooved grain. The tests with the smooth bore grain were conducted first and the results used to make modifications to the grooved grain prior to its use.

The first test was conducted without the use of hydrogen as a pilot flame fuel. Ignition did not occur, even though the ignitor was observed to be functioning normally when the video tape was reviewed. This result confirmed the expected behavior based on earlier testing.

In test number two the hydrogen was used, but combustion was not attained due to the ignition limits at the combustor inlet. The static combustion chamber pressure (P_c), stagnation pressure (P_t), and the static wall pressure (P_w) in the vicinity of the stagnation probe, when used in the equation 2 indicated that shockdown or choking had occurred due to mixing without heat addition.

In test number three, the hydrogen flow rate was increased and sustained combustion was attained. A hydrogen mass flow rate approximately 0.08 percent of that of the air flow proved to be sufficient. For initial evaluation, the combustion process of the hydrogen in the grain was assumed to be a Rayleigh-line flow and a comparison of the stagnation

temperature ratio required to thermally choke the flow and the theoretical ratio in the motor with only the hydrogen burning was made. By entering PCPEP with the amount of hydrogen required to achieve ignition, it was observed that the heat addition due to the hydrogen alone, as predicted by T_t theoretical and T_{ti} when used in the equation

$$\frac{T_{t2}}{T_{t1}} = \frac{f(M_1)}{f(M_2)}$$

where

$$f(M) = \frac{1+[(\gamma-1)/2]M^2}{(1+\gamma M^2)^2} M^2,$$

was sufficient to thermally choke the flow from the Mach 1.5 inlet condition. Thus, in the initial configuration, both heat addition and mixing losses resulted in shockdown/choking of the flow.

The fourth test was conducted under the same conditions as the third test, except that, due to the burning of some of the Plexiglas fuel in the previous test, the initial bore diameter was 0.68 in. instead of 0.562. Pressure tap locations are shown in Figure 10 and the measured data are presented in Table 1. Upon ignition, the peak pressures at P_w and P_t, when substituted into equation 2, indicate that the initial Mach number was approximately 1.06 at the end of the fuel grain. Complete shockdown did occur, as expected, with

SMOOTH BORE GRAIN DATA

	P_c psia	P_w psia	P_t psia	T_i °R	ΔW_t lbm	t_b sec	m_i lb/s	m_f lb/s	M_e	r in/s
RUN 1	65	65	62	1370	---	---	.453	---	---	---
RUN 2	112	63	110	1150	---	---	.474	---	.931	---
RUN 4 (at ign)	133	57	115	1100	---	---	.459	.06	1.1	---
RUN 4 (8 sec)	60	40	27*	1300	---	8	.459	.06	.80*	---
RUN 4 (shtdwn)	20	22	15*	1350	.566	11.56	.459	.06	--	.038

* Pitot failure had occurred.

Table 1. Smooth bore grain test data.

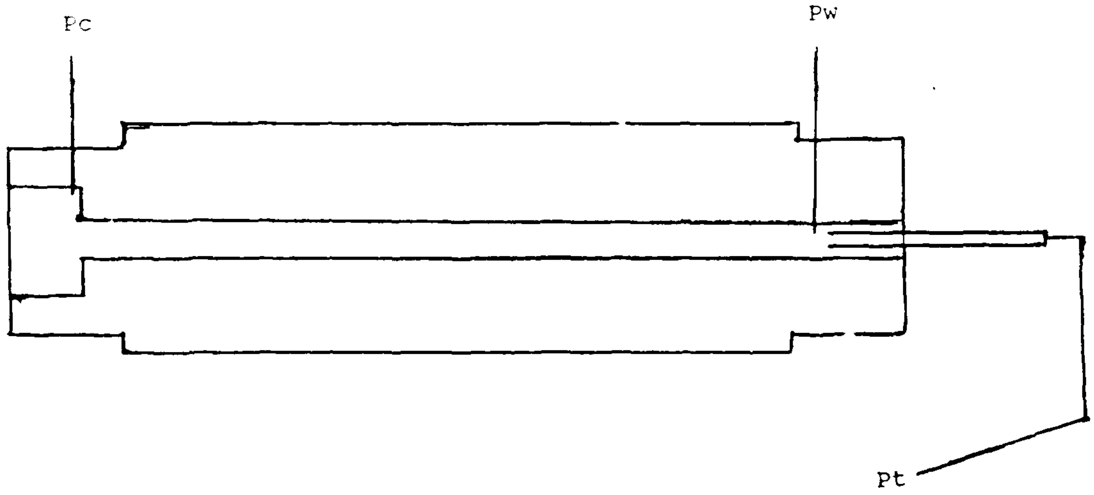


Figure 10 - Pressure tap locations on grain 1.

the shock location beginning at the exit of the grain and progressing toward the head end as the bore expanded. The shockdown was evidenced by the burning patterns in the grain and the results of inserting measured pressures into equation 2. The normal shock location in the grain stabilized at a position approximately six inches from the beginning of the bore, and turbulent mixing and combustion occurred from that position to the end of the grain as evidenced by the burn pattern in the grain surface.

The average regression rate was measured to be approximately 0.038 in/sec, which was very close to the rate of 0.035 in/sec predicted using the following regression rate formula for subsonic applications:

$$r = 2.3 \times 10^{-4} P^{.51} T^{.34} G^{.41}.$$

It was not possible to determine whether the regression rate was constant through the test, or whether the rate increased as the bore area increased.

The video tape replay (Appendix C) indicated that the Plexiglas was burning from the head end through the bore. A blue flame, indicating the combustion of a lean mixture, was evident from the head of the grain to a position several inches beyond the exit. The uncooled steel probe used to measure the stagnation pressure turned white hot and melted away during the run, indicating temperatures well beyond the

$1484^{o}R$ predicted temperature generated by PCPEP using the air and heater gas and hydrogen flow rates measured during the test. The theoretical combustion temperature with the inclusion of the total fuel mass flow rate was $4422^{o}R$. When only 60 percent of the fuel mass flow rate was used (the portion attributable to the bore upstream of the shocks), the estimated temperature was $3643^{o}R$. The destruction of the probe was not likely to occur if only the hydrogen were burning.

Early in the test, the pressure ratios indicated that the flow had shocked down or had been thermally choked as in the earlier tests. At the end of the test, the area ratio of the grain bore to inlet nozzle was 10.37, which, assuming a γ of 1.3, yields an inlet Mach number of 3.6. Using the Rayleigh temperature ratio expression again with that Mach number and γ, indicated that in order to thermally choke the flow the stagnation temperature must be approximately $5129^{o}R$ This was above the maximum possible combustion temperature and therefore indicated that both heat addition and friction/mixing losses contributed to the observed shockdown.

2. Grooved Grain Test Results

Two tests were conducted using the grooved grain, which had been modified as shown in Figure 11 to take advantage of the findings in the smooth bore tests. Since the shock

pattern appeared to stabilize at a distance of six inches from the end of the smooth bore grain, the diameter of the bore of the grooved grain was expanded to two inches in an attempt to prevent or delay shock formation.

The motor ignition was very fast, and the fuel regression rate was measured to be twice that of the smooth bore. There appeared to be some burning of the Plexiglas, as evidenced again by the presence of the blue flame through the entire length of the combustor.

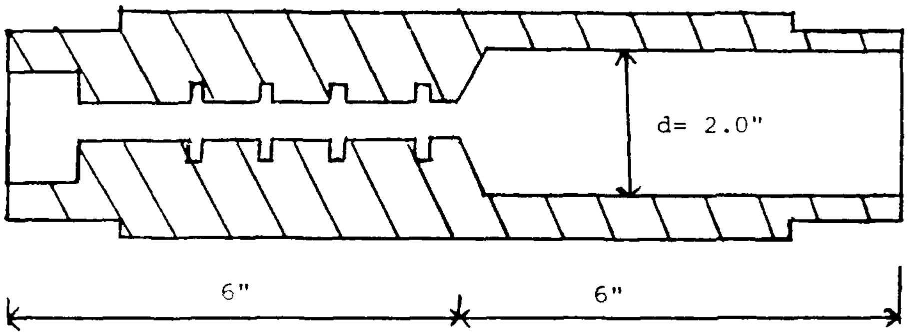


Figure 11 - Schematic of scramjet modified fuel grain.

As in the smooth bore, there was complete shockdown prior to ignition. This occurred even though the grain diameter was enlarged midway along the grain. Thus, the grooves in the fuel surface significantly improved ignition and fuel regression rate, but also increased the mixing/heat addition losses. After ignition, the reduction of pressures as the bore cross sectional area increased indicated that the flow was either shocked down or choked, but lack of P_t data

or visual evidence prevented determination of the shock location. The grooves, and the burning pattern they created, prevented visual determination of the location, since there were no smooth walls to compare and find the turbulent burn pattern or bulges on the wall. Although a water cooled stagnation probe was used, the data collected was inconsistant, and, therefore not used.

Due to the inability to determine the shock location, it was not possible to conclude whether or not supersonic combustion occurred. Pressure tap locations for the grooved grain are shown in Figure 12, and the measured data are displayed in Table 2.

Photographs of the first test are given in Appendix C.

Attempts to gain ignition on the second test were unsuccessful even though all conditions were the same as for the first test with the exception of the bore diameter. Because of the previous burn, the bore diameter had expanded to one inch from the original 0.562 in. Thus, the bore entry Mach number was approximately 2.90 instead of the original Mach 1.6. Although the system was burning strongly when the extinguishing purge was initiated for the first run, the grain could not be re-ignited under the same mass flow conditions. This showed that for the solid fueled scramjet, ignition limits may be considerably more restrictive than flammability limits.

GROOVED GRAIN TEST DATA

	AIR ONLY	IGNITION PEAK	IGNITION OFF	SHUT DOWN	
P_c	120	185	72	52	psia
P_w	60	93	45	29	psia
P_c/P_w	2.0	1.99	1.60	1.79	
M_w	1.06	1.06	.86	.97	
ΔW_t				.365	lbm
t_b				7	sec
r				.068	in/sec

Table 2. Grooved grain test data.

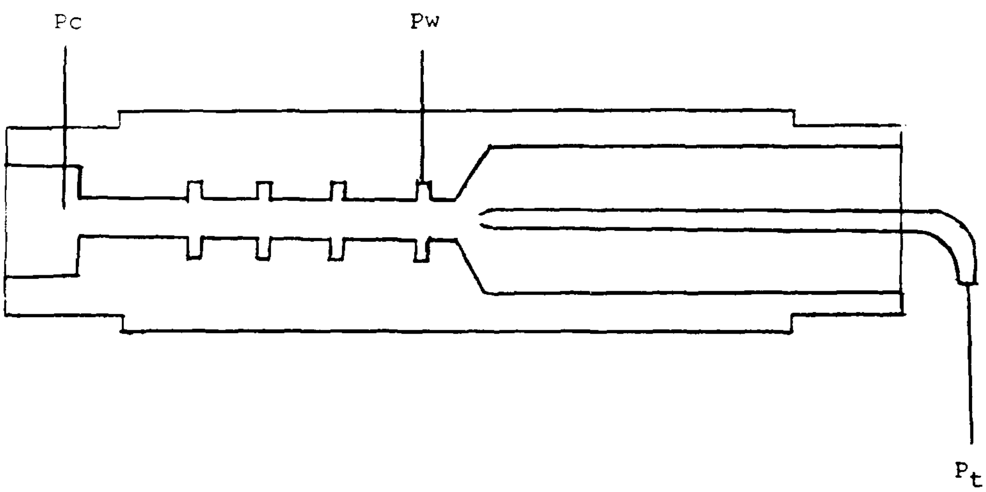


Figure 12 - Pressure tap locations on scramjet grain 2.

B. DMRJ RESULTS

Two test runs of the DMRJ were completed. With the exception of the locations of the data collection points, the conditions were the same for both runs. In both cases flames and smoke were observed at the exit of the combustor tube. During the second test, low frequency (approximatey 50 Hz) chugs were heard, indicating flame instability. The only difference in the flow path between tests was the removal of the thermocouples in the SFRJ exhaust and in the mixing areas for the second test, which may have affected the mixing and flame stability. Pressure tap locations are depicted in Figure 13, and the data collected are shown in Table 3.

DMRJ TEST DATA

	P_c psia	P_{mix} psia	T_{mix} °R	P_1 psia	M_{mix}	PHI SFRJ	PHI DMRJ	T_{bp} °R
RUN 1 AIR ONLY	55	9.8	1080	13.9	1.78	---	---	1113
AFTER IGN.	105	3.7	>2700	16.6	2.78	1.89	.893	1113

	P_c	P_{mix}	P_1	P_2	P_3	M_{mix}	T_{bp}	PHI_{DMRJ}
RUN 2 AIR ONLY	55	9.8	9.4	12.1	13.7	1.78	1112	---
AFTER IGN.	108	2.8	11.3	9.1	16.6	2.85	1112	.893

Table 3. DMRJ test data.

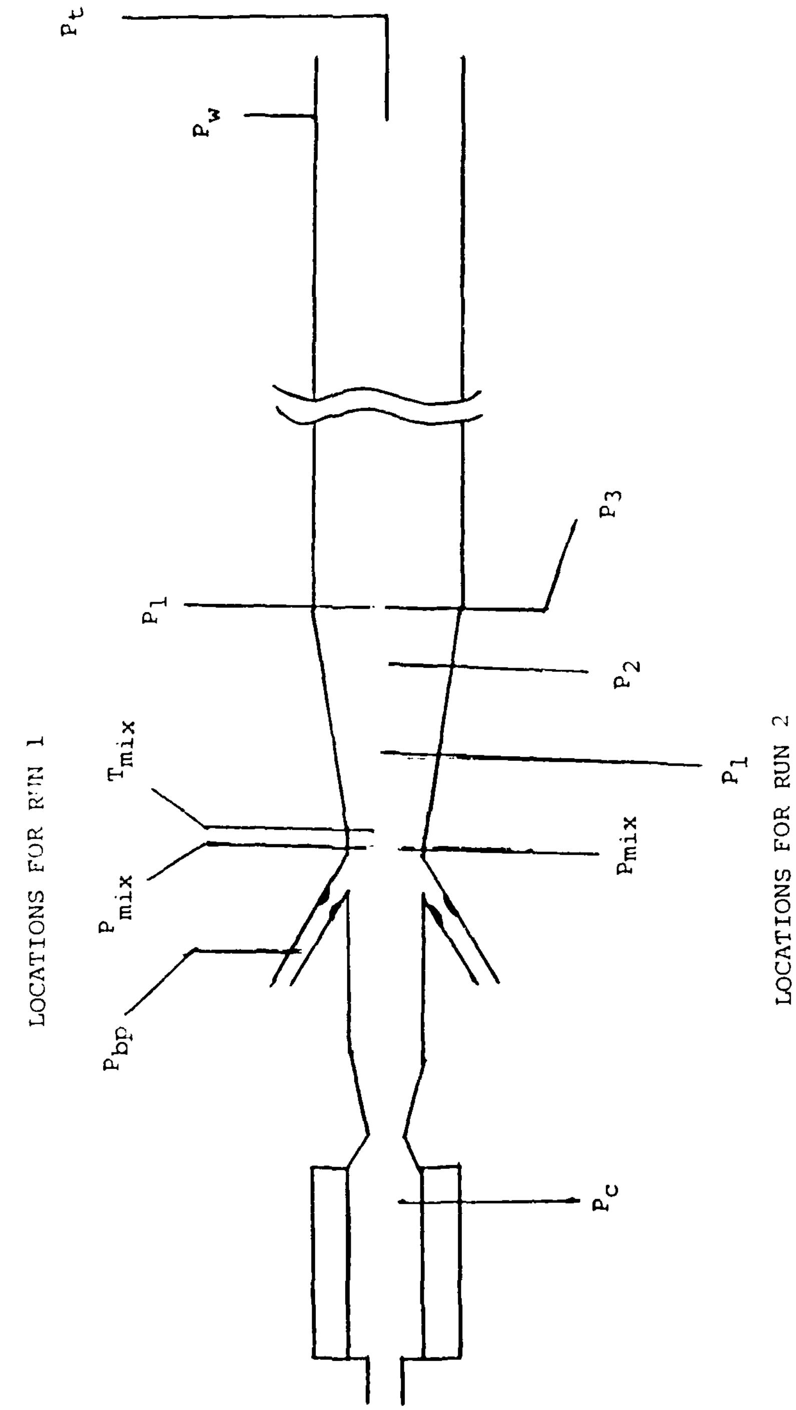


Figure 13 - Pressure and temperature locations for DMRJ.

When only heated air was run through the motor before ignition, the Mach number at p_{mix}, using the pressure relationship

$$P_t = p\left(1 + \frac{\gamma - 1}{2}M^2\right)^{\frac{\gamma}{\gamma - 1}} ,$$

was Mach 1.78. This value is somewhat lower than expected for an area ratio of 4.38, which would normally indicate Mach 3.04. There may have been some flow separation during the expansion that might have caused this anomaly. Nonetheless, the flow at the entrance to the mixing region was supersonic. Between p_{mix} and p_1 the static pressure rose from 9.8 psia to 13.9 psia, indicating that the mixing losses, without heat addition, were nearly sufficient to cause complete shockdown even though the flow was expanding through the region. It was expected that the 30° dump angle for the bypass air would cause significant mixing/shock losses, but these losses were also coupled with the friction losses downstream in the constant area pipe.

After ignition the pressure ratio at p_{mix} indicates a Mach number of 2.78. Between p_{mix} and p_1 there was a pressure rise to 16.6 psia, indicating that a shockdown occurred in the expansion zone from the combined effects of mixing and heat addition along the entire length of the burner.

If there was no combustion beyond the SFRJ exhaust, the expected value of the temperature at T_{mix} would be

approximately as indicated by averaging the mass flow rates at the expected temperatures in the following expression:

$$T_{mix_{th}} = \frac{(m_a * T_c + m_{abp} * T_{bp})}{m_a + m_{abp}} = 2365^{\circ}R.$$

The actual value was greater than 2700°R. The thermocouple was destroyed during the test. This high temperature indicated that spontaneous combustion occurred in the supersonic region prior to the p_1 position.

In the second test the conditions without heat addition were nearly identical to the first. The pressure at p_{mix} was slightly lower, indicating a small rise in the Mach number, but the flow was returned to the same conditions as the first test by the time it reached the end of the expansion zone. For this test the pressure tap locations were adjusted to attempt to locate the shock position. The pressure rises between p_1 and p_2 and p_2 and p_3 would seem to indicate a series of shocks in the mixing and combustion areas resulting in complete shockdown prior to entry into the large diameter constant area tube.

P_t data was unavailable for either test due to the destruction of the probe during the runs, so performance data in terms of thermal efficiency was not available. The original intent of the experimental investigation was to build the supersonic combustor with excessive length to ensure time

for combustion. If combustion was difficult to attain, a diverging section would have increased the Mach number and made the combustion environment even more difficult. Thus, the initial configuration used a long constant area section downstream of the initial diverging section. Once spontaneous combustion was attained the length of the combustor was to be shortened in increments, each time measuring the combustion efficiency and stagnation pressure losses. Due to unplanned delays in fabrication and experimental difficulties only the initial configuration was evaluated. The data indicate that supersonic combustion was probably occurring and subsequent efforts should be made with a shorter constant area section or continuously diverging combustor walls.

V. CONCLUSIONS AND RECOMMENDATIONS

The experimental results indicated that ramjet combustion in supersonic flows can be initiated and maintained using selected solid fuels in both the dual mode and scramjet configurations. The hypothesis that the chemical reactions are rapid enough to sustain combustion in combustors of reasonable length has been experimentally supported. Based on the observations, the limiting factor to improving the performance of engines of this type to the point of practical application is the geometry of the grain and the combustor. In this investigation the losses due to mixing and configuration initiated shocks accounted for the majority of the possible causes for the failure or poor performance of the system.

The differences between ignition and flammability limits was found to be critical in the case of the solid fuel scramjet. The motor may have to be ignited at low supersonic or subsonic speeds. The erosion of the grain will cause an increase in the velocity due to the increasing area.

The use of a cooled and shielded pressure probe for the gathering of stagnation pressure data is highly recommended. Unshielded, uncooled probes are not sufficiently robust to perform satisfactorily under the conditions experienced.

Varying the grain lengths and expansion rates to minimize shockdown should be investigated to find the optimum performance. Experiments involving the determination of the minimum required hydrogen pilot fuel and the ignition/flammability limits of both combustor types should be conducted.

With regard to the high mixing losses experienced in the DMRJ, coaxial injection of the bypass air into the flow of the DMRJ may cause fewer mixing losses and still provide adequate mixing. The use of non-concentric nozzles in the SFRJ portion of the DMRJ may also provide enhanced mixing characteristics.

Based on the initial results obtained in this investigation, it appears that spontaneous combustion of the fuel rich SFRJ exhaust did occur. A systematic investigation should now be made to determine the optimum combustor configuration for minimizing the mixing and friction losses.

LIST OF REFERENCES

1. Billig, F.S., Tactical Missile Design Concepts, Vol. 4, Johns Hopkins University, APL Technical Digest, Nov. 3, 1983.

2. Netzer, D. W., Overview of Tactical Missile Propulsion Design, Naval Postgraduate School, Monterey California.

3. United Technologies, Chemical Systems Division, The Pocket Ramjet Reader, 1978, p.28.

4. Avery, W.H., Twenty-Five Years of Ramjet Development, Vol. 25, No. 11, Jet Propulsion, Nov. 1955.

5. Schetz, J. A., Billig, F. S., and Favin, S., "Analysis of Mixing and Combustion in a Scramjet Combustor with a Co-Axial Fuel Jet", paper AIAA-80-1256, AIAA/SAE/ASME 16th Joint Propulsion Conference,1980.

6. Vaught, C.B., Witt, M.A., Netzer, D.W.,and Gany, A., "Feasibility Study of Using Solid Fuels for High Mach Number Tactical Ramjets", paper in preparation, June 1989.

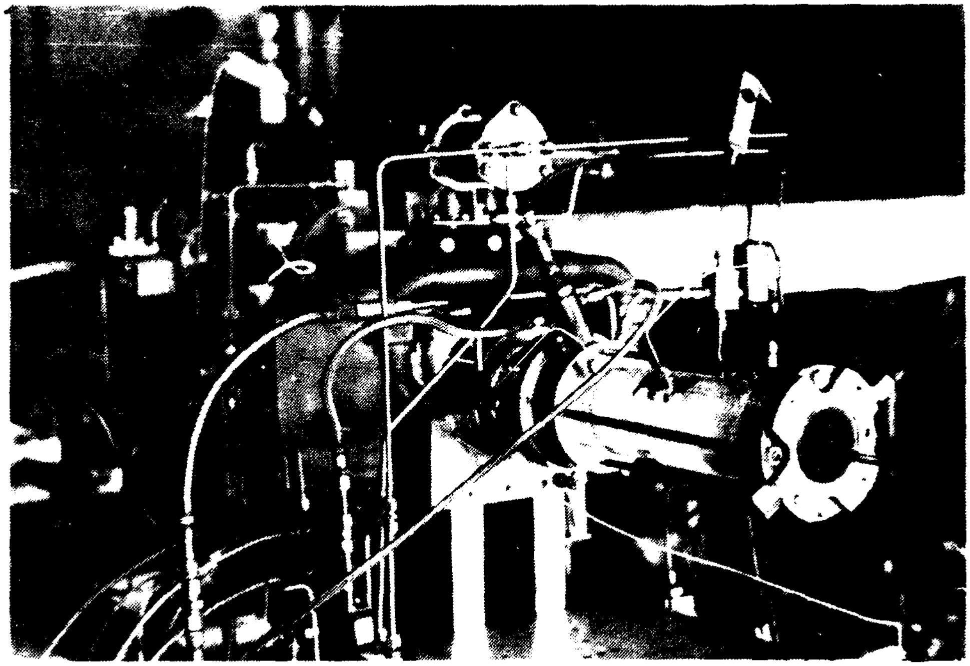
7. Flow Research, Inc., NASP Contractor Report 1028, Shock Waves for Enhanced Mixing in Scramjet Combustors, by S. Menon, p. 2, December 1988.

APPENDIX A

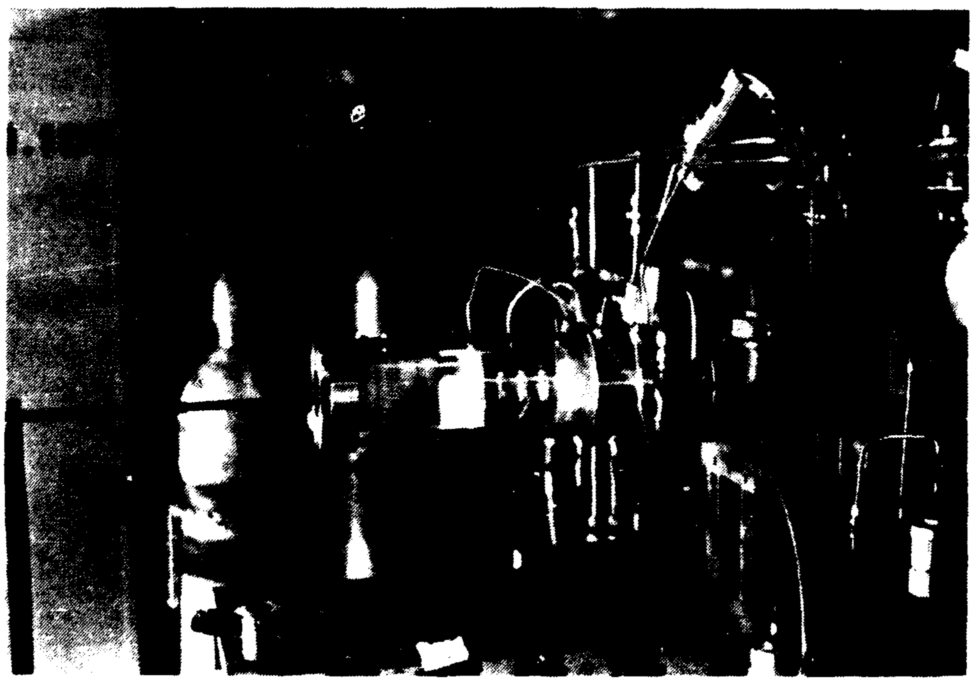
```
PEPOUT.DAT                6/20/89          Page 1
1                          **** NEWPEP - April 1988 ****
 * air/h2              * 06/20/89  * DH ** DENS **** COMPOSITION *******

 AIR (750 K) (1350 R)                113  0.00001  835N 2240   5AR
 HYDROGEN (GASEOUS)                    0  0.00001    2H

 INGREDIENT WEIGHTS (IN ORDER) AND TOTAL WEIGHT        (LAST ITEM IN LIST)

    99.9100      0.0900   100.0000

 THE PROPELLANT DENSITY IS 0.00001 LB/CU-IN OR  0.0003 GM/CC

 NUMBER OF GRAM ATOMS OF EACH ELEMENT PRESENT IN INGREDIENTS

    0.089286 H     5.389063 N     1.445689 O     0.032270 AR

 ****************************CHAMBER RESULTS FOLLOW     ************************

  T(K)  T(F)  P(ATM)   P(PSI) ENTHALPY  ENTROPY   CP/CV    GAS    RT/V
  847. 1065.    6.80   100.00    11.29   178.15  1.3484  3.472    1.959   TCRE

 DAMPED AND UNDAMPED SPEED OF SOUND= 1883.891 AND 1883.893 FT/SEC

 SPECIFIC HEAT (MOLAR) OF GAS AND TOTAL=    7.691     7.691
 NUMBER MOLS GAS AND CONDENSED=   3.4719    0.0000

   2.69450 N2          0.70049 O2          0.04463 H2O         0.03227 Ar
  1.67E-05 NO         7.99E-06 NO2

 THE MOLECULAR WEIGHT OF THE MIXTURE IS   28.802

 TOTAL HEAT CONTENT (298 REF)   = 143.415 CAL/GM
 SENSIBLE HEAT CONTENT (298 REF)= 138.689 CAL/GM
```

```
PEPOUT.DAT              6/20/89          Page 1
1                          **** NEWPEP - April 1988 ****
 * PMM/H2,AIR          * 06/20/89  * DH ** DENS **** COMPOSITION *******

 AIR (750 K) (1350 R)                  113  0.00001  835N 2240    5AR
 HYDROGEN (GASEOUS)                      0  0.00001    2H
 PLEXIGLASS                           -906  0.04260    8H   5C    2O

 INGREDIENT WEIGHTS (IN ORDER) AND TOTAL WEIGHT          (LAST ITEM IN LIST)

    90.3130     0.0870     9.6000   100.0000

 THE PROPELLANT DENSITY IS 0.00001 LB/CU-IN OR  0.0003 GM/CC

 NUMBER OF GRAM ATOMS OF EACH ELEMENT PRESENT IN INGREDIENTS

    0.853397 H      0.479430 C      4.871409 N      1.498593 O
    0.029170 AR

 ****************************CHAMBER RESULTS FOLLOW     *************************

  T(K)  T(F)  P(ATM)    P(PSI) ENTHALPY  ENTROPY   CP/CV     GAS    RT/V
 2456. 3962.    6.80    100.00     1.51   215.98  1.2491  3.455     1.969    TCRE

 DAMPED AND UNDAMPED SPEED OF SOUND= 3079.565 AND 3079.569 FT/SEC

 SPECIFIC HEAT (MOLAR) OF GAS AND TOTAL=    9.963    9.963
 NUMBER MOLS GAS AND CONDENSED=   3.4545   0.0000

   2.42502 N2          0.44277 CO2         0.41233 H2O         0.06220 O2
   0.03661 CO          0.02917 Ar          0.02131 NO          0.01621 HO
  5.67E-03 H2         2.07E-03 O          1.11E-03 H          2.07E-05 HO2
  1.38E-05 NO2        2.61E-06 N2O

 THE MOLECULAR WEIGHT OF THE MIXTURE IS   28.947

 TOTAL HEAT CONTENT (298 REF)    = 716.700 CAL/GM
 SENSIBLE HEAT CONTENT (298 REF)= 671.597 CAL/GM


 *                     * 06/21/89  * DH ** DENS **** COMPOSITION *******

 AIR (700 K)                           101  0.00001  835N 2240    5AR
 HTPB (SINCLAIR)                        13  0.03320  103H  73C    1O

 INGREDIENT WEIGHTS (IN ORDER) AND TOTAL WEIGHT          (LAST ITEM IN LIST)

    87.8100    12.1900   100.0000

 THE PROPELLANT DENSITY IS 0.00001 LB/CU-IN OR  0.0003 GM/CC

 NUMBER OF GRAM ATOMS OF EACH ELEMENT PRESENT IN INGREDIENTS

    1.259819 H      0.892882 C      4.736399 N      1.282834 O
    0.028362 AR

 ****************************CHAMBER RESULTS FOLLOW     *************************

  T(K)  T(F)  P(ATM)    P(PSI) ENTHALPY  ENTROPY   CP/CV     GAS    PT/V
 2054. 3238.    7.82    115.00     9.03   228.01  1.2833  3.920     1.996    TCRE

 DAMPED AND UNDAMPED SPEED OF SOUND= 3040.694 AND 3040.697 FT/SEC

 SPECIFIC HEAT (MOLAR) OF GAS AND TOTAL=    9.001    9.001
 NUMBER MOLS GAS AND CONDENSED=   3.9198   0.0000

   2.36817 N2          0.77176 CO          0.36054 H2          0.26879 H2O
   0.12107 CO2         0.02836 Ar          0.00099 H           0.00008 HO
  1.25E-05 NH3        8.05E-06 NO         3.27E-06 CNH

 THE MOLECULAR WEIGHT OF THE MIXTURE IS   25.511

 TOTAL HEAT CONTENT (298 REF)    = 591.418 CAL/GM
 SENSIBLE HEAT CONTENT (298 REF)= 562.848 CAL/GM
```

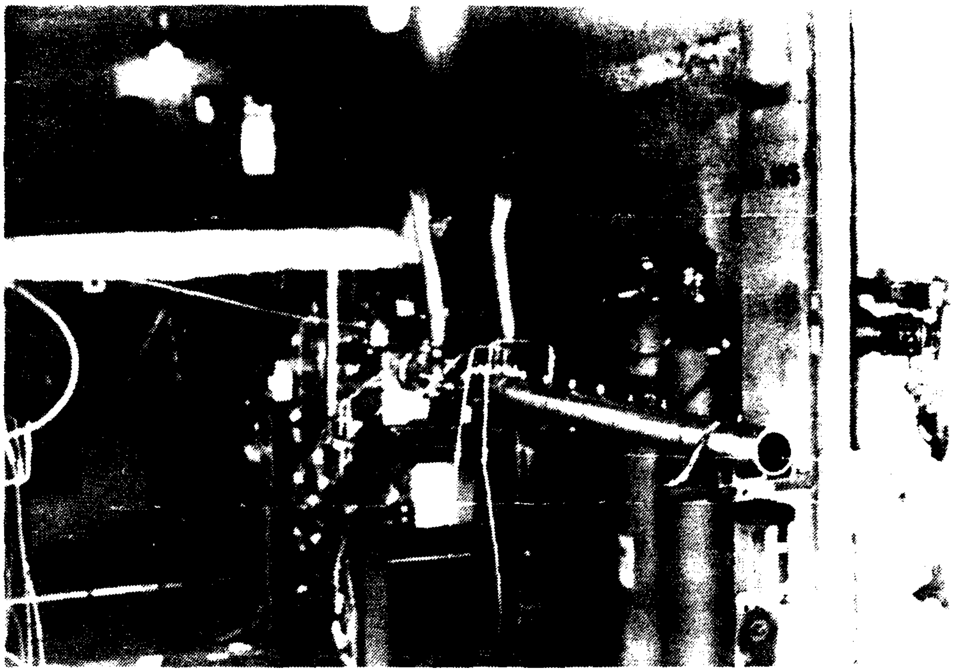
APPENDIX B



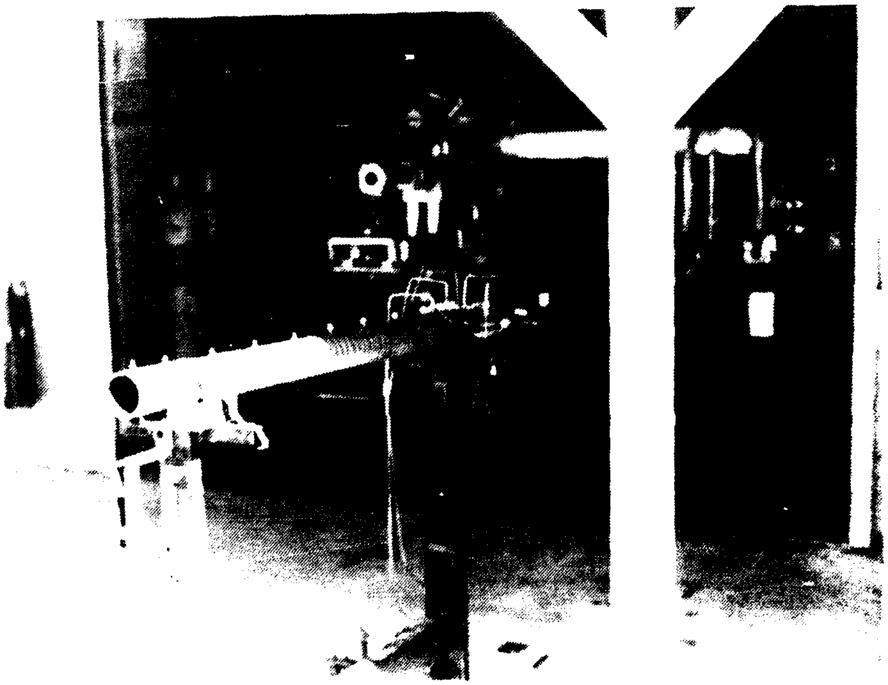
(a) Scramjet test stand



(b) Scramjet test stand with grooved grain

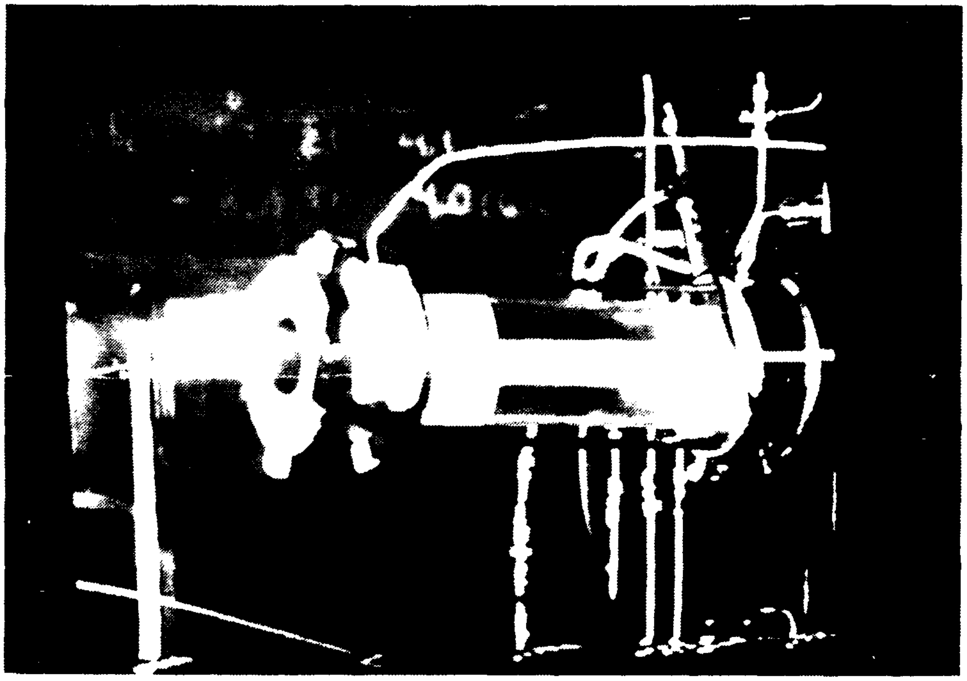


(c) Dual mode test stand

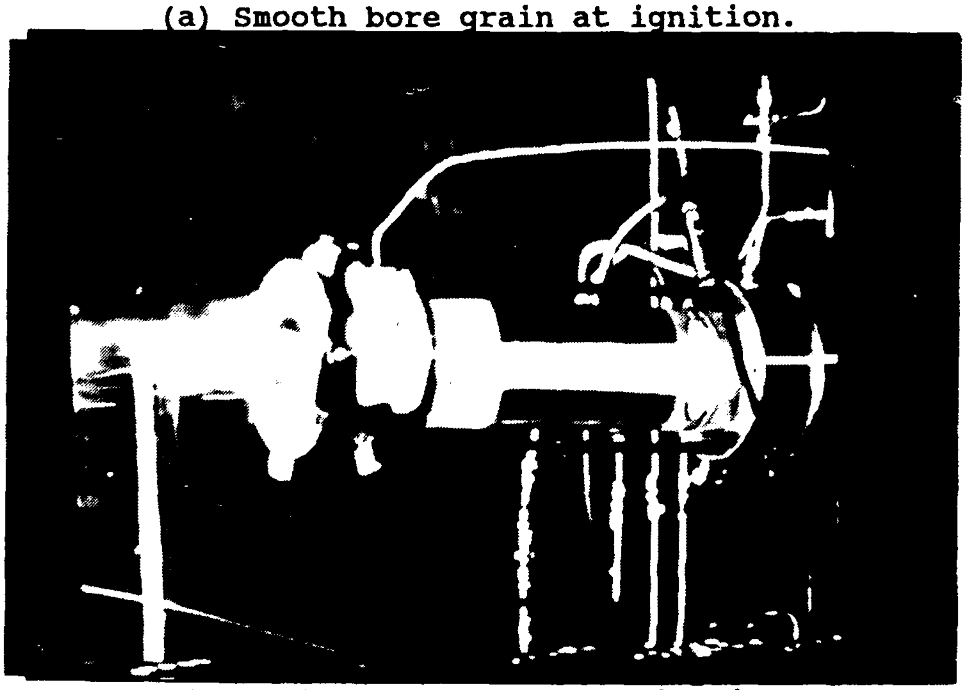


(d) Dual mode supersonic combustor tube

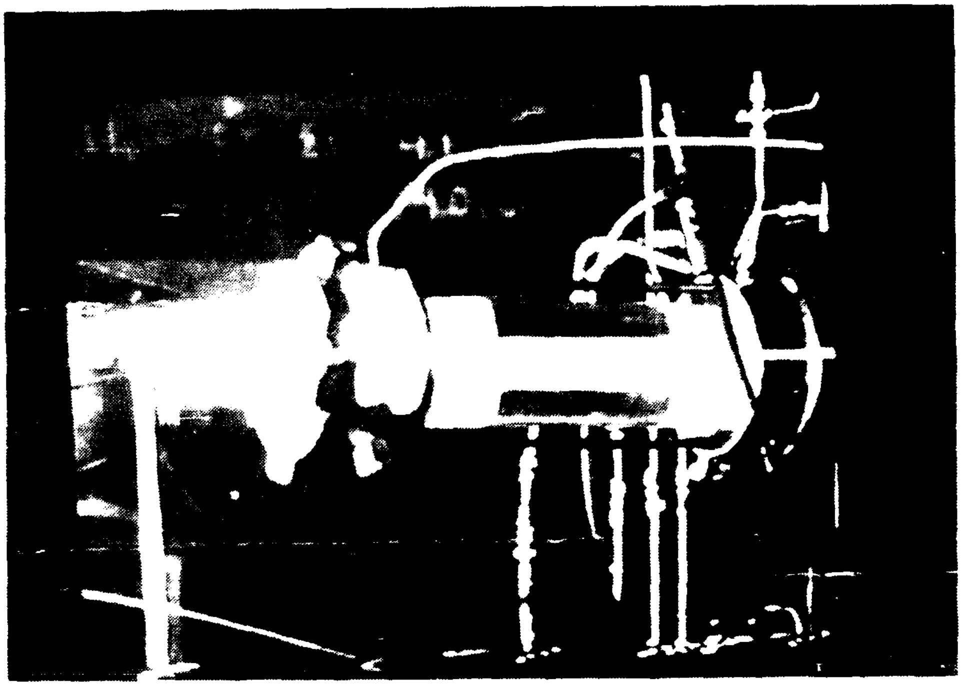
APPENDIX C



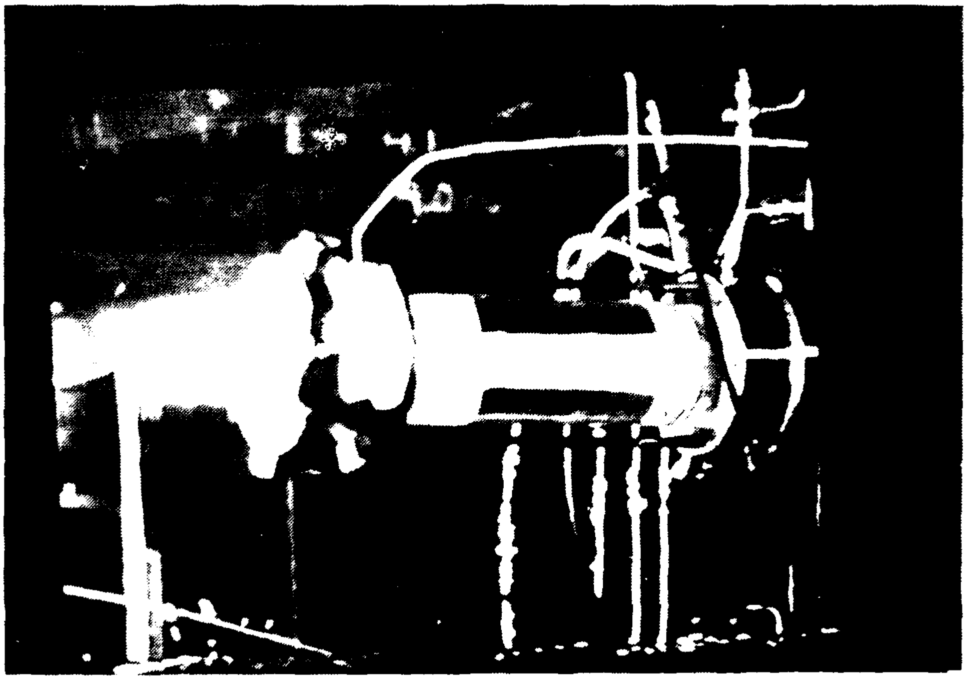
(a) Smooth bore grain at ignition.



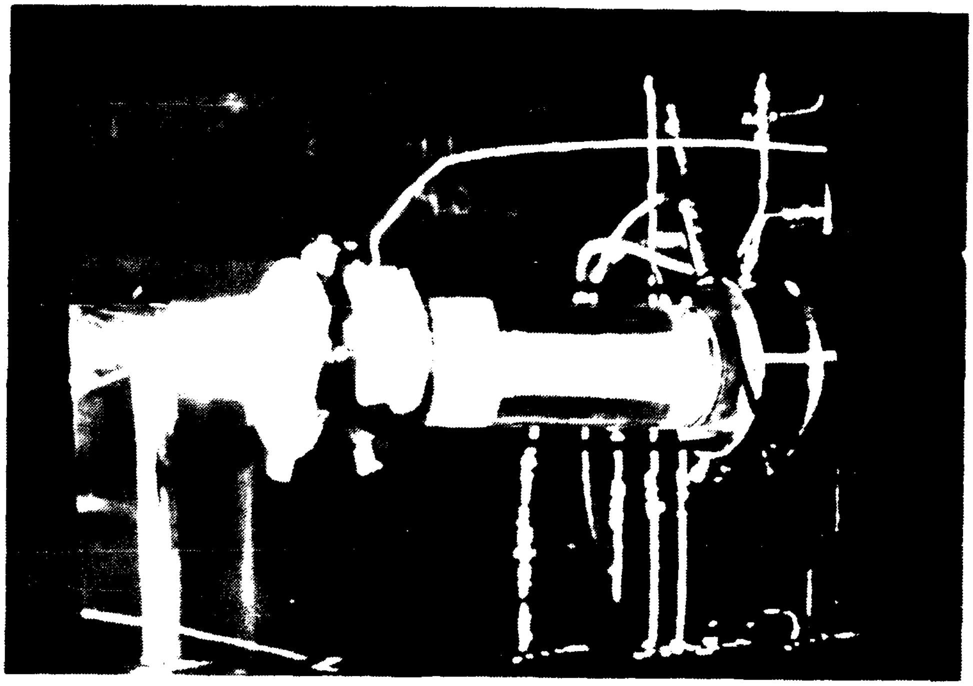
(b) First signs of bore wall rippling as shock moves up grain.



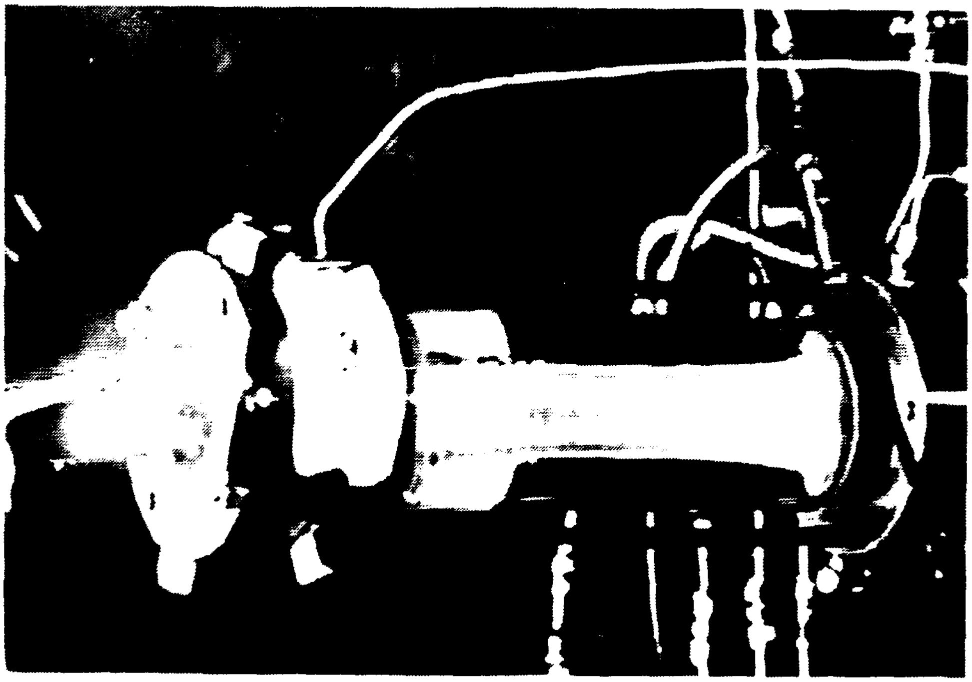
(c) Probe white hot. Rippling more pronounced and continuing forward.



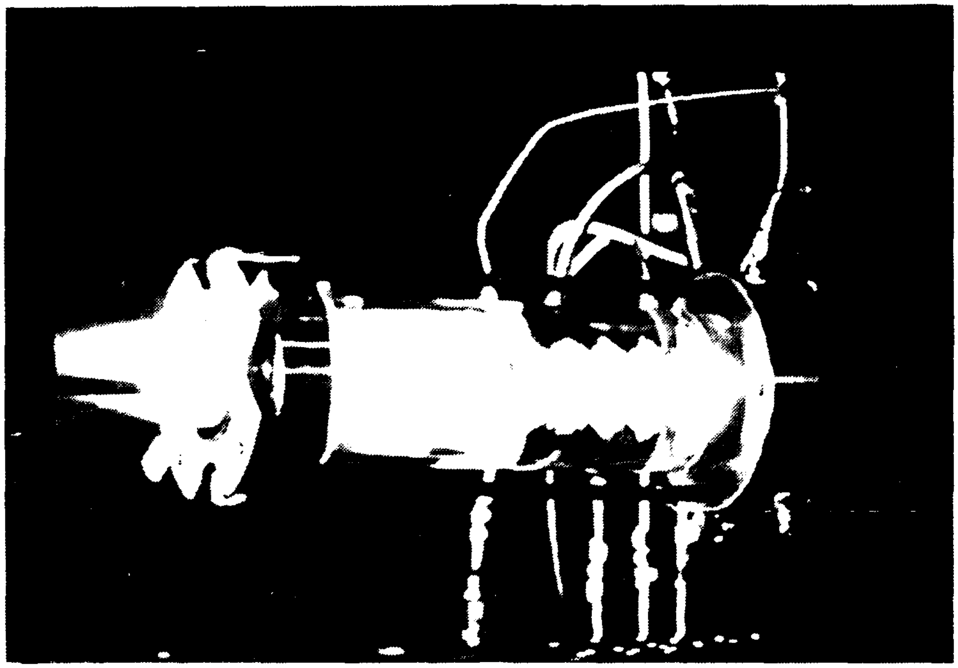
(d) Probe melting. Bore widening with shock approaching furthest point of advance.



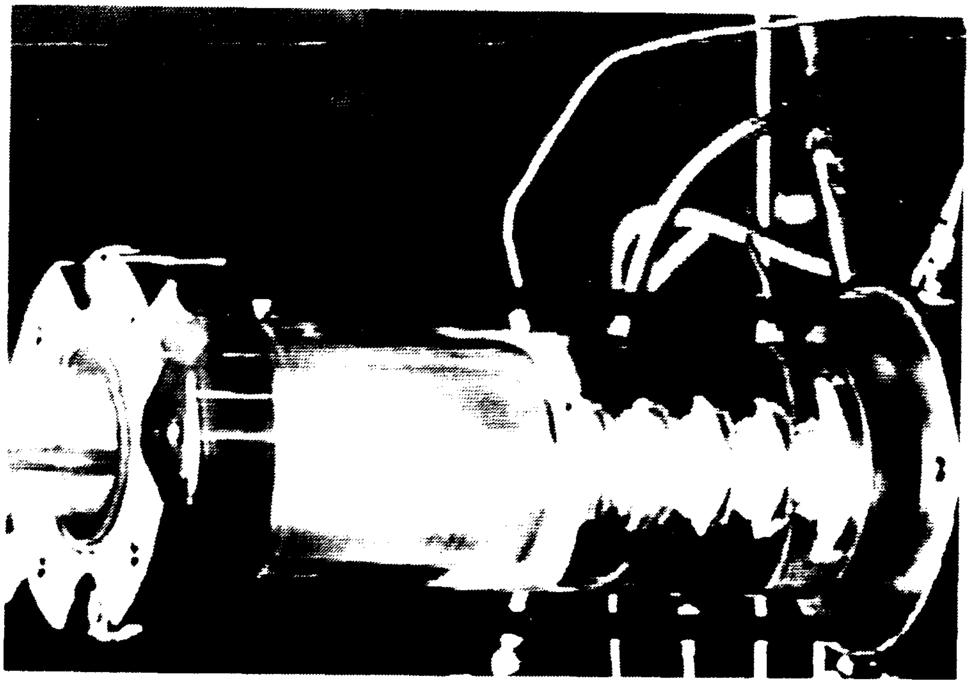
(e) Probe ejected from exit. Bore widening with little or no shock advance.



(f) Close up of bore burn pattern near end of run.



(g) Grooved grain near ignition.



(h) Grooved grain near end of run.

INITIAL DISTRIBUTION LIST

1.	Defense Technical Information Center Cameron Station Alexandria, Virginia 22304-6145	1
2.	Naval Weapons Center China Lake, California 93555-6001 F. Zarlingo, Code 32403	1
3.	Library, Code 0142 Naval Postgraduate School Monterey, California 93943-5002	1
4.	Department Chairman Department of Aeronautics, Code 67 Naval Postgraduate School Monterey, California 93943-5026	1
5.	Prof. D. W. Netzer Department of Aeronautics, Code 67Nt Naval Postgraduate School Monterey, California 93943-5004	4
6.	Prof. R. Shreeve Department of Aeronautics, Code 67Sf Naval Postgraduate School Monterey, California 93943-5004	1
7.	Dr. Alon Gany Department of Aeronautical Engineering Technion, Haifa 32000, Isreal	1
8.	Lcdr. Michael A. Witt Executive Officer USS Taylor (FFG 50) FPO Miami 34093	6